Appendix B
Propulsion Primer, Performance Parameters, and Units

Since man first began to fly, he has striven to go faster. His progress in pushing speed records upwards was followed by an upward trend for commercial aircraft (fig. 61). As speed of aircraft increases, so does drag—the frictional resistance on its surfaces. It follows that higher speeds require higher thrust to overcome drag (fig. 62).

The Propulsion Principle

The thrust of all flight propulsion systems comes from the same principle—reaction—as expressed by Newton's second law of motion:

$$\text{force} = \text{mass} \cdot \text{acceleration}$$

In the case of airplanes flying through air, boats moving across the water, or rockets flying in the atmosphere or outside it, the reaction principle can be expressed in the equivalent forms:

$$\text{force} = \text{rate of change of momentum, or}$$
$$\text{force} = \text{mass flow rate} \cdot \text{change in fluid velocity.}$$

The airscrew or propeller, the boat propeller, the jet engine that swallows atmospheric air in operation, and the rocket engine that carries its oxidizer as well as fuel, all use the reaction principle. If you have stood at the stern of a boat under power, for example, and observed the wake, you have seen that the wake moves faster than the adjacent water, which is a visual indication of the change in fluid velocity.

Flight propulsion systems vary in the relative amounts of mass flow rates and changes in fluid velocity, as illustrated diagrammatically by figure 63. Assume, for comparison purposes, that the three types of vehicles are moving horizontally through the atmosphere at the same flight speed, designated by V_o, and that all are producing the same propulsive thrust, designated as F. Let us now examine, on a relative basis, the amount of fluid mass affected and the velocity given to it. On this relative basis, the propeller affects the largest air mass but gives it the lowest increase in velocity. The

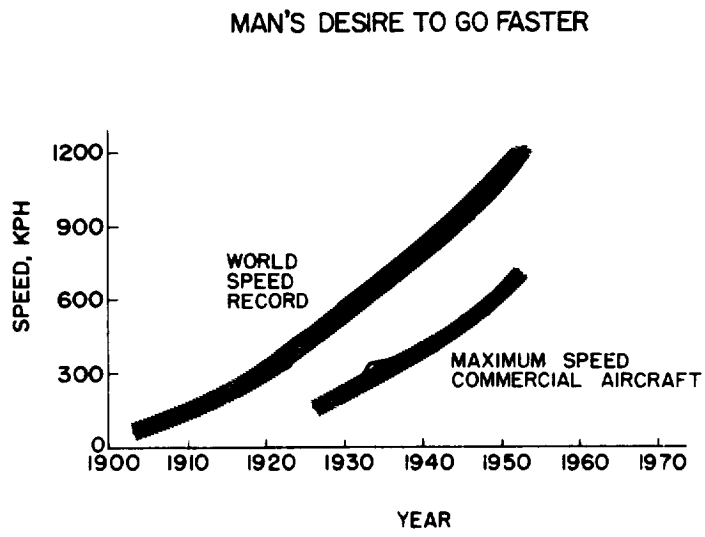

Fig. 61. Growth of aircraft speed. (Adapted from NACA figure, 1953.)

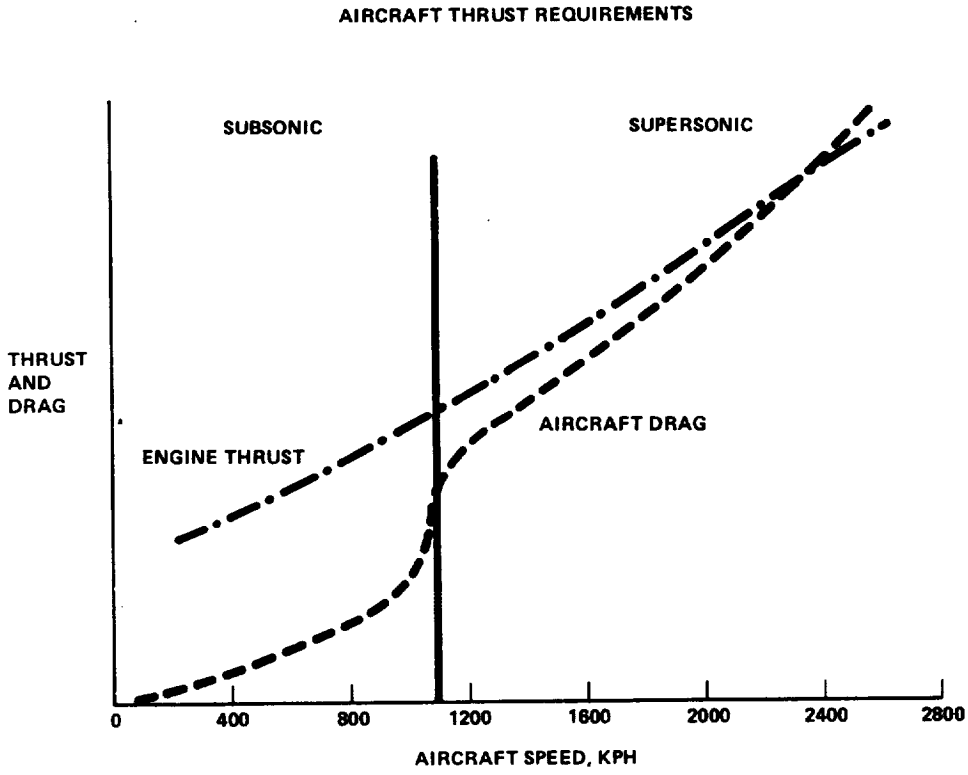

Fig. 62. Aircraft thrust requirements increase with speed. (Adapted from NACA figure, 1953.)

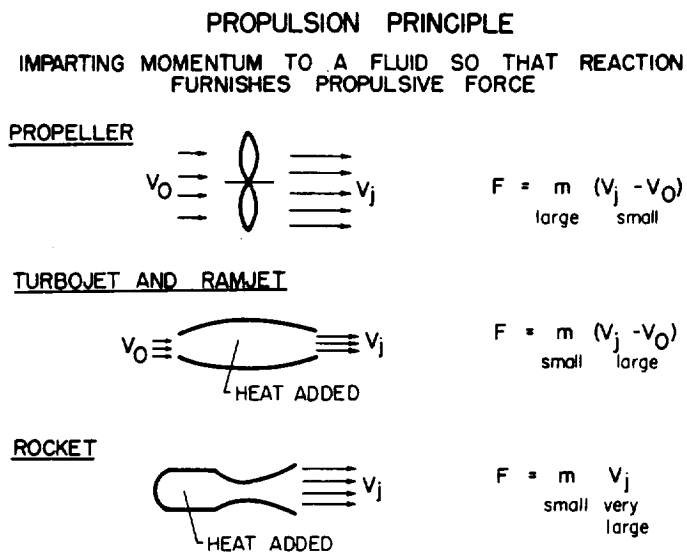

Fig. 63 Propulsion principle is the same for propeller, turbojet, ramjet, and rocket. F = thrust (assumed same for all in comparison); m = mass flow of working fluid (air or exhaust gas); V_j = exhaust gas velocity; V_0 = flight velocity. (NACA figure, 1953.)

turbojet and ramjet engines affect a smaller mass of air than the propeller, but give the air a much higher velocity. The rocket, carrying its own working fluid as fuel and oxidizer, does not accelerate air but for the same thrust, ejects the smallest amount of fluid mass at the highest velocity.

Since both the propeller and air-consuming jet engines must accelerate the air they fly through to produce thrust, it follows that their thrust depends upon flight velocity. For the propeller, the limitation begins to show when its tip approaches the speed of sound in the air (340 m/s at sea level; 295 m/s at 10700 m altitude). In the mid-1930s, an airplane speed of 805 km/hr (224 m/s) was attainable only in power dives and there was uncertainty over whether full power on the engine speeded up the dive or slowed it down. For the air-consuming jet engine, the momentum drag of the incoming air increases with flight speed and when flight speed equals the exhaust jet speed, thrust falls to zero. The rocket, on the other hand, carries all of its working fluid and its jet thrust is independent of flight speed, in or out of the atmosphere. These relationships are illustrated qualitatively by figure 64.

Propulsive Efficiency

To go deeper into the subject of propulsive effectiveness, we can consider the propulsive efficiency. Marine engineers call it the Froude efficiency after William Froude (1810–1879) who first used it. It is defined as the ratio of useful power output to the rate of energy input. For flight, propulsive efficiency is expressed as:

$$\frac{\text{thrust} \cdot \text{flight speed}}{\text{kinetic energy increase of fluid}}$$

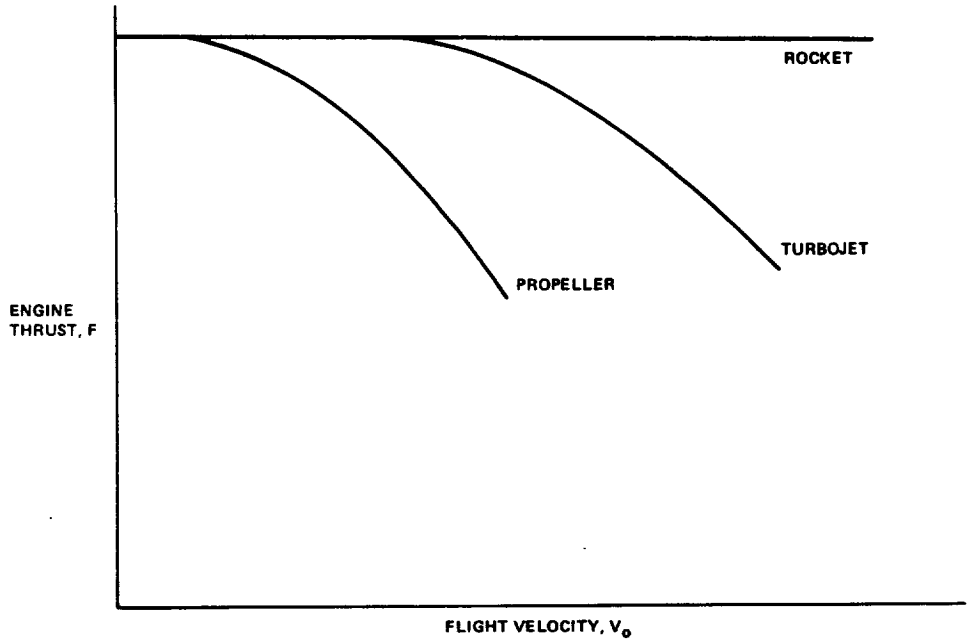

Fig. 64. Rocket thrust is independent of flight velocity, while the turbojet and propeller depend upon it.

Its principal use for flight is to indicate that the various propulsion systems operate best in different speed ranges. This is shown qualitatively by figure 65.* The propeller is the most efficient propulsive method at low speeds, while the jet engine achieves best efficiency only at relatively high flight speeds. The very high exhaust velocities of the rocket make its propulsive efficiency high only at very high flight speeds. Figure 66, from an NACA figure of the 1950s, shows similar information in a slightly different way.

Propulsion System Comparisons

The considerations of thrust and propulsive efficiency are by no means the whole story in comparing various propulsion systems. We must consider, for example, the thermodynamic efficiency of the engine cycle, or how efficiently the chemical energy of the fuel is converted to useful work such as driving the propeller and compressor, or accelerating the exhaust jet. The weight and size of the engine, its complexity, and its service life are other factors to be considered. Figure 67, from an NACA chart before

*Since thrust is the mass flow rate change in fluid velocity, propulsive efficiency is

$$\frac{m\,(V_j - V_o\,V_o)}{1/2\,m\,(V_j - V_o)\,V_o}, \text{ which reduces to } \frac{2}{1 + V_j/\,V_0}$$

where m is mass flow rate, V_j is exhaust jet velocity, and V_o is flight velocity.

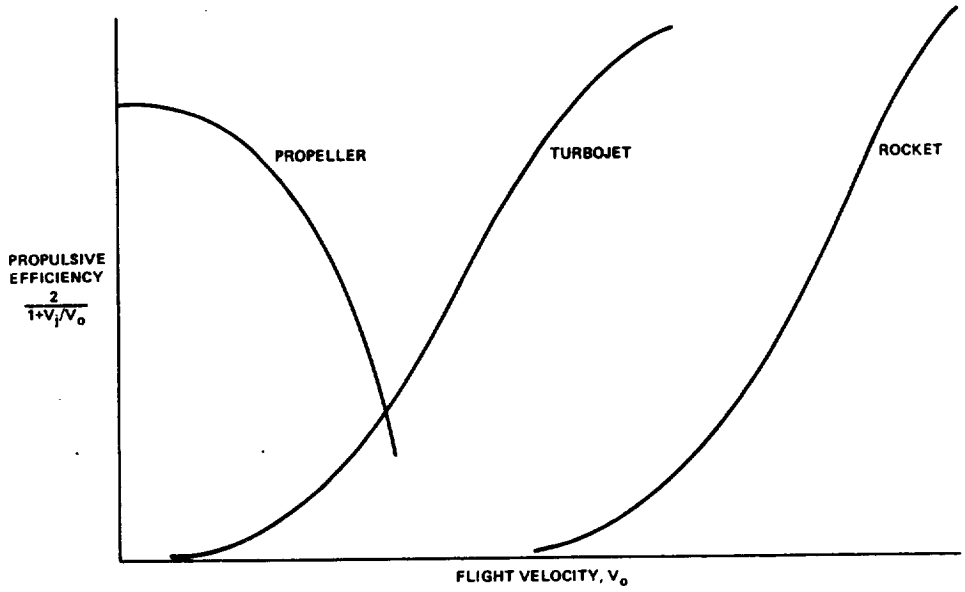

Fig. 65. Propulsive efficiency peaks at different speeds for the propeller, turbojet, and rocket.

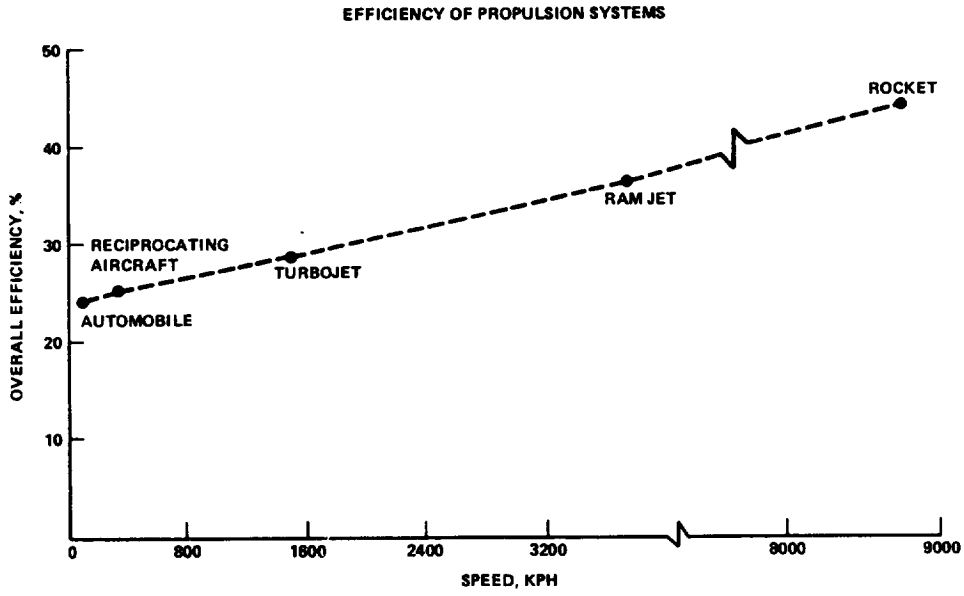

Fig. 66. Efficiency of various propulsion systems peaks at different speeds. (Adapted from NACA figure, ca. 1953.)

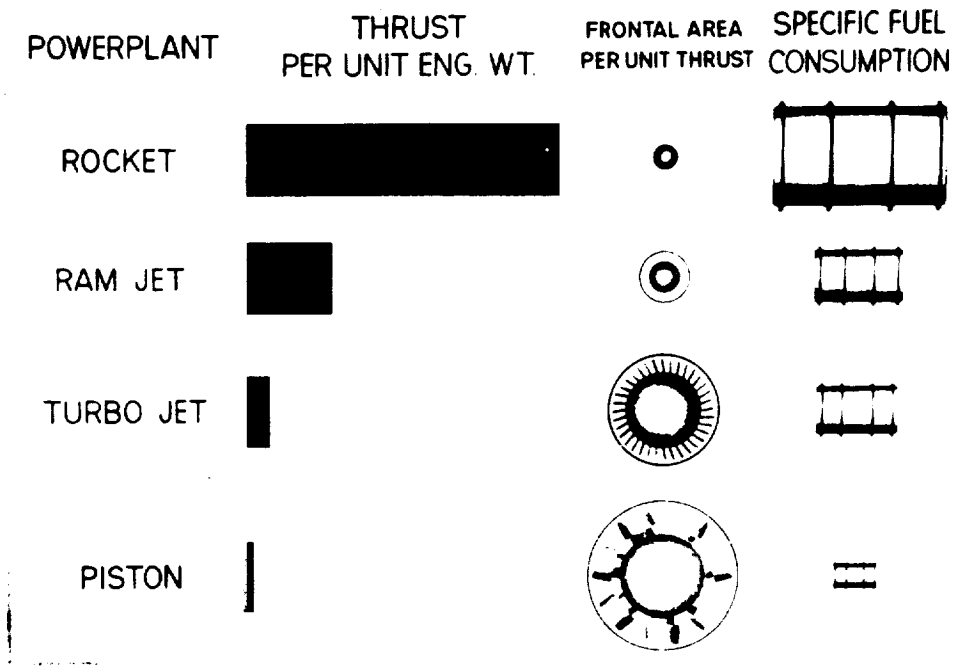

Fig. 67. Comparison of flight propulsion systems; relative values. (NACA figure, ca. 1949.)

1950, compares flight engines on the basis of specific weight, frontal area (which affects drag), and specific fuel consumption.

Gas Turbine Engines

The gas turbine is much older than the piston (reciprocating) engine. Hero used a gas turbine (steam) to drive a toy merry-go-round in 130 B.C. A patent for a gas turbine was granted to John Barber in England in 1791, and many were developed for various applications in the nineteenth century. Jets of hot gas impinge against turbine blades to spin the turbine which, in turn, performs some useful function such as driving a generator, compressor, pump, or other mechanical device. In an aircraft turbine engine, the turbine drives a compressor, fan, and sometimes a propeller, depending on the engine design. The most usual turbine engine is the turbojet where the turbine drives a compressor. Figure 68 is a sectional view of a turbojet, with the various components labeled. Air entering the inlet is compressed by the axial flow compressor having multiple stages. A liquid is sometimes injected in the compressor of high-performance turbojets as a coolant. The main fuel is injected and burned in the combustors in the annular space between the outer shell and the inner "spool." The hot combustion gases drive the turbine, shown in two stages, and in the after section additional fuel may be injected and burned to increase gas temperature and provide more thrust. The hot exhaust gases expand through the nozzle to produce thrust.

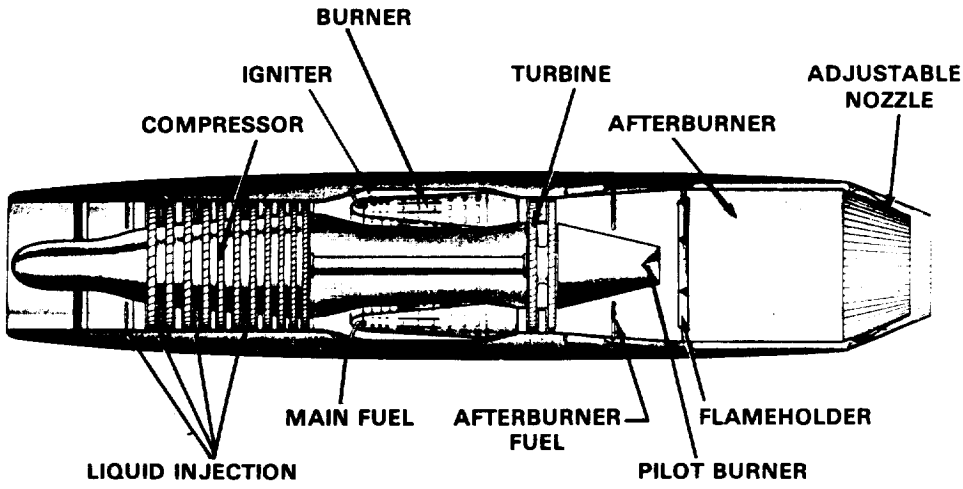

Fig. 68. Turbojet engine with thrust augmentation by an afterburner. (NACA figure, 1952.)

Thrust augmentation with an afterburner is for high performance military turbojets; afterburners are not used on commercial turbojets, where low specific fuel consumption is a major factor. Figure 69 shows the growth of thrust of turbojet engines from the early 1940s until 1952.

Ramjet

The ramjet engine began to receive attention during the second half of the 1940s and reached its peak during the 1950s. The ramjet has been called a flying stovepipe, for the absence of rotating parts that characterize the turbine engine. The ramjet gets its name from the method of air compression; it cannot operate from a standing start but must first be accelerated to a high speed by another means of propulsion. The air enters the spike-shaped inlet and diffuser (fig. 70) which serve the same purpose as the compressor. Fuel is injected and burns with the aid of flameholders that stabilize the flame. The burning fuel imparts thermal energy to the gas, and the expansion through the nozzle at speeds greater than the entering air produces the forward thrust. The ramjet, always needing an auxiliary propulsion system for starting, got squeezed between improved turbine engines and rockets during the 1950s and never recovered.

Rocket

A typical liquid propellant rocket, as diagrammed by NACA in 1952, is shown by figure 71. It consists of a guidance compartment, payload, fuel tank, oxidant tank, and engine compartment. A gas generator, operated from the main propellants or an auxiliary propellant, drives a turbine which drives pumps to supply fuel and oxidant at pressures of 20 to 40 atmospheres to the thrust chamber. The fuel is usually circulated in a cooling jacket surrounding the nozzle and combustion chamber prior to injection

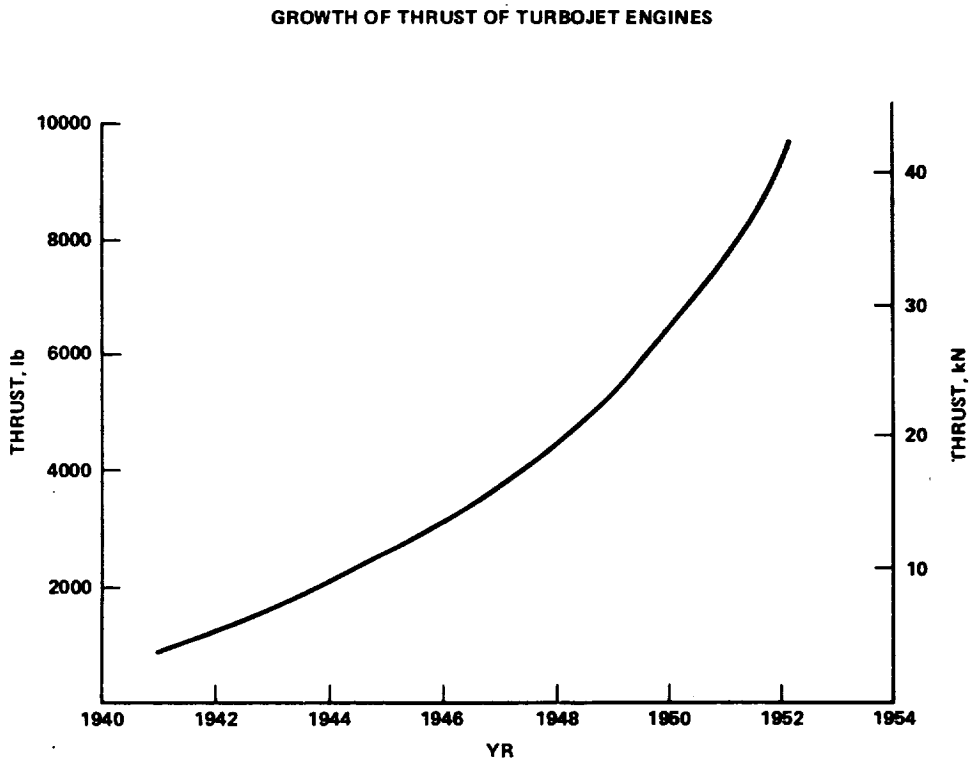

Fig. 69. Growth of thrust of turbojet engines. (Adapted from NACA figure, 1953.)

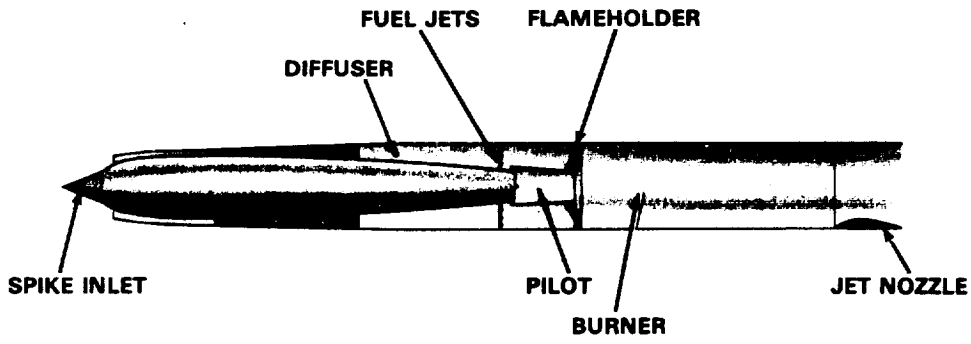

Fig. 70. A typical ramjet engine. (NACA figure, 1952.)

and burning. This is called regenerative cooling, for the heat picked up by the coolant-fuel is returned to the combustion chamber during combustion. Some fuel-oxidant combinations react spontaneously on initial contact—aniline and nitric acid, for example, as well as all combinations that use fluorine as the oxidizer. Other combinations, such as gasoline-oxygen, alcohol-oxygen, and hydrogen-oxygen,

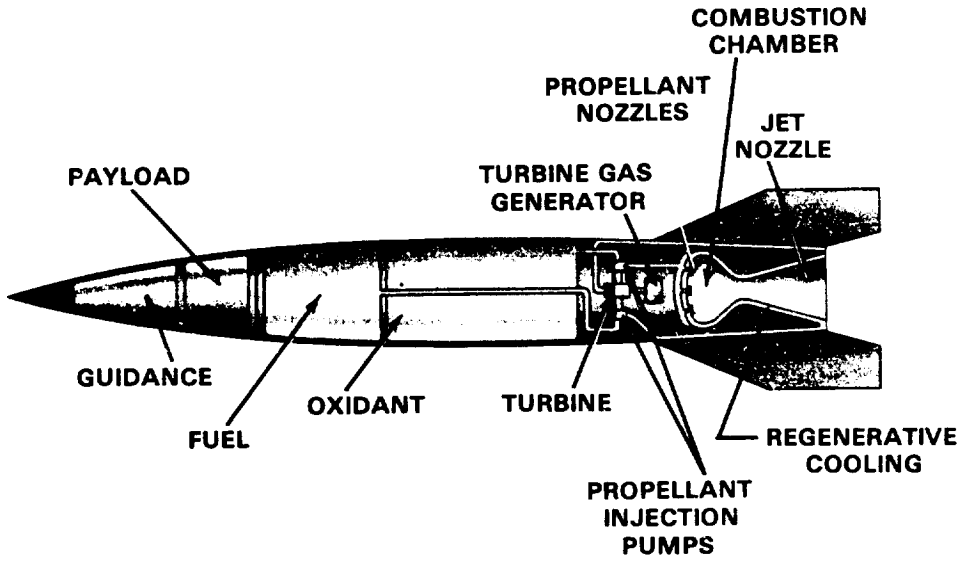

Fig. 71. A typical liquid-propellant rocket. (NACA figure, 1952.)

require an ignition source, which may be electrical or chemical. The fuel and oxidizer burn in the combustion chamber, and the hot gases expand through the nozzle to produce forward thrust. Because they use concentrated oxidizers, rockets are more compact than air-consuming engines where the oxidizer is gaseous, and diluted 80 percent by nitrogen at that.

Rocket Performance Parameters and Units

American rocket research and development during 1945–1959 used the English system of units. As NASA has directed that metric units be used in all publications, including this one, the conventional units have been converted using E. A. Mechtly, "The International System of Units: Physical Constants and Conversion Factors," (NASA SP-7012, 1973). The international system, designated SI in all languages, mandates newtons for force, newtons per square meter for pressure, and joules per second per square meter for heat transfer rates. In the interest of readability and as a concession to those brought up in the English system, a few compromises have been made. These are described in the following discussion of the major performance parameters used in rocketry.

Thrust. Force, expressed in newtons (N), kilonewtons (kN), and meganewtons (MN). A newton is that force that gives 1 kilogram (kg) of mass an acceleration of 1 meter per second per second (m/s^2). To convert from 1 pound force to newtons, multiply by 4.448. This has usually been rounded off to 4.45 for conversions in the text. To offset the unfamiliarity of the newton, thrust is normally expressed in pounds in parentheses.

Propellant flow. Expressed in kilograms per second (kg/s). 1 lb = 0.4536 kg.

Mixture ratio. The proportions in which fuel and oxidizer are burned in the combustion chamber. Mixture ratio is expressed in several ways; one of the most common is the ratio of mass flow of oxidizer to mass flow of fuel, abbreviated O/F; sometimes the inverse is used. Another way of expressing mixture ratio, popular in the 1950s, is the percentage of the total mass flow that is fuel; a third is the molar ratio of one to the other. A mole is the mass equivalent of the molecular weight of the fuel and oxidizer. Sometimes a stoichiometric mixture ratio is mentioned; this means the exact proportion of fuel and oxidizer for complete combustion. As an example,

$$H_2 + {}^1/_2O_2 \rightarrow H_2O$$

is the stoichiometric mixture. In this example, one mole of hydrogen combines with a half mole of oxygen to produce a mole of water. Expressed in mass units, 2.016 grams of hydrogen plus 16 grams of oxygen produce 18.016 grams of water. The O/F is 7.9, percent fuel is 11, and molar oxidizer to fuel ratio is ½.

Specific impulse and exhaust velocity. In 1903 Tsiolkovskiy, and other Europeans after him, expressed rocket engine performance in terms of the velocity of the exhaust emerging from the nozzle in meters per second (m/s). This made sense because the rocket exhaust velocity was a term in the equation expressing the velocity of a rocket-propelled vehicle. In the United States, it became the custom to express rocket performance in terms of the measured quantities: thrust and mass flow of the propellants. The thrust divided by the total mass flow of propellant was defined as the specific impulse. Specific impulse is the inverse of specific fuel consumption used in discussing the performance of other types of propulsion systems. In English units, specific impulse is in pounds force per pounds mass per second (lbf·sec/lbm). On seeing pounds in both numerator and denominator, many succumbed to the temptation to cancel them and express specific impulse incorrectly in units of seconds: the two pounds represent different physical phenomena, force and mass, and are connected by the conversion factor 32.2 lbm·ft/lbm·sec^2. In SI, specific impulse is expressed in newtons per kilogram per second or N·s/kg. English values of specific impulse are converted to SI by multiplying by 9.807, which can be rounded to 10 for approximations. The numerical value of specific impulse and exhaust velocity in SI are the same; only the units are different. Since exhaust velocity is a simple concept to visualize physically and since specific impulse expressed in newtons per kilogram per second is unfamiliar to many, including the author, all performance values in this text have been converted to exhaust velocity in meters/second (m/s). Typical values of exhaust velocity for liquid propellant rockets range from 2000 to 4500 m/s. The V-2 had an exhaust velocity of about 2200 m/s, very good for 1944. High energy propellants give exhaust velocities in the range of 3000 to 4500 m/s, and the liquid hydrogen–oxygen combination is in the upper part of this range.

Pressure. Expressed as newtons per square meter (N/m^2) in SI. One pound per square inch is 6895 N/m^2. Rocket combustion pressures used during 1945–1959 were generally in the range of 300–600 lb/in^2 or 2069–4137 kN/m^2. Rocket combustion

pressures were also expressed in atmospheres—multiples of sea-level pressure which is the same in any system of units. Since atmospheres are easier grasped than kN/m^2, combustion and higher pressures in this text are expressed in atmospheres (atm). One atmosphere is slightly over 100 kN/m^2; the 2069–4137 kN/m^2 pressure range above becomes 20.4–40.7 atm.

Nozzle area ratio. The ratio of nozzle exit area to throat area. Area ratio determines the amount of expansion of the exhaust gases through the nozzle and is related to exhaust gas pressures. If a rocket designer is asked to provide a nozzle that is to be operated only on the ground—as in the case of an experimental rocket engine—he chooses an area ratio such that the exhaust gas pressure at the nozzle exit is equal to ambient pressure, usually considered as sea-level pressure or 1 atmosphere. If the combustion pressure is 20 atmospheres, the gases undergo an expansion ratio of 20 and this corresponds roughly to a nozzle area ratio of 4. If, for some reason, the designer provides a nozzle area less than that needed for complete expansion, the exhaust gases emerge from the nozzle exit at greater than ambient pressure. In this case the gases are said to be underexpanded, for they have to expand further to reach ambient pressure. On the other hand, if the designer provides a larger area ratio than that needed for complete expansion, the exhaust gases reach a pressure equal to ambient while still in the nozzle. In some cases the gases will continue to follow the nozzle walls and expand to a pressure lower than ambient. In this case the gases are said to be overexpanded. Sooner or later, overexpanded gas must be reconciled with the ambient pressure and nature provides for this adjustment by means of a shock wave. The ideal nozzle is one that provides for complete gas expansion, neither more nor less, for theory shows that this yields maximum exhaust velocity. This poses a problem to the designer of a launch vehicle, because as soon as the vehicle is launched, the ambient pressure begins to fall, approaching zero at very high altitudes. Since the nozzle is fixed, what area ratio should he choose? If he designs for sea-level conditions, he is getting less performance at altitude than could be realized; if he designs for altitude, he suffers some performance loss from overexpansion at sea-level. For rocket engines in upper stages, he doesn't have this problem for the ambient pressure is close to zero at ignition. In rocket engines for all stages, however, he must balance the gain in performance from providing a larger area ratio nozzle against the added weight and cooling requirements of the larger nozzle and must also stay within special limits of the vehicle. A typical rocket engine for an upper stage will have an area ratio of about 40. In the text, you will seldom encounter area ratio, but performance at sea-level and altitude is used and in each case it is implied that a nozzle is provided for complete, or nearly complete, expansion of the exhaust gases.

Heat transfer rate. Heat flow per unit time per unit area. In SI, joules/second·meter^2 ($J/s \cdot m^2$). One $Btu/s \cdot in^2$ equals 1.637 $J/s \cdot m^2$. Rocket combustion temperatures, from 2500 to 4500 K, and pressures, from 20 to 40 atmospheres, produce very high heat transfer rates—on the order of 20 to 200 times greater than those produced in boilers and superheaters in a steam plant, for example, and much higher than in other types of internal combustion engines. Typical values of rocket heat transfer rates range from 1 to 20 $J/s \cdot m^2$, but can be higher in local areas.

Mass ratio. The ratio of the gross (total) mass of a rocket vehicle to its empty mass (M_o/M_e). The difference between the two is the mass of propellant expended during the operation. The empty mass includes the rocket engine, the structure, controls, and the payload. For the last stage of a multistage launch vehicle, the payload is the spacecraft. For other stages, the payload is the loaded stages above it. Mass ratios range from about 3 to 10. The V-2 had a mass ratio of about 3.

Vehicle velocity. Expressed in meters per second (m/s). Vehicle velocity is a function of the rocket exhaust gas velocity, mass ratio of the vehicle, aerodynamic drag during flight through the atmosphere, gravitational pull expressed in terms of burning time of the rocket, and the trajectory. In 1903 Tsiolkovskiy derived the velocity for a rocket in vertical flight, disregarding drag and gravitational pull. This yielded the fundamental rocket flight equation named after him and so identified in the text. The equation is:

$$V = V_e \ln(M_o/M_e)$$

where V is vehicle velocity, V_e is exhaust gas velocity, M_o is gross mass, M_e is empty mass, and ln is the natural logarithm. The direct relationship between vehicle velocity and exhaust gas velocity accounts for the interest in high energy propellants that yield high exhaust velocities.

Multistage rocket. A vehicle composed of two or more complete rocket systems, each including propellant and associated controls and structure. The last (upper) stage carries the spacecraft as the payload. The payload of the first stage is the upper stages and the spacecraft. Investigators learned a very long time ago that rocket vehicle velocity could be increased by having one rocket unit riding piggy-back on another until the first expends all of its propellant. The dead weight of the first stage is normally jettisoned. The second rocket unit or stage begins operating at the velocity given it by the first stage and adds its own velocity increase, for a much higher final velocity. If the first two stages have identical exhaust velocities and mass ratios, for example, the final velocity will be twice that of either operating singly.

Other parameters and units. Horsepower is given in its equivalent kilowatts (kW), and revolutions per minute (rpm) was retained. The conversion of feet to meters and pounds to kilograms is obvious.

TABLE 14.—*Propulsion Parameters and Units*

Parameter	English	SI	This text
Thrust	lb	N	N (lb)
Propellant flow	lbm/s	kg/s	kg/s
Mixture ratio	various		various
Specific impulse	lbf·s/lbm	N·s/kg	not used
Exhaust velocity	ft/s	m/s	m/s
Pressure	lbf/in^2	N/m^2	kN/m^2 or atm
Nozzle area ratio	A_e/A_t		A_e/A_t
Heat transfer rate	$Btu/s \cdot in^2$	$J/s \cdot m^2$	$J/s \cdot m^2$
Mass ratio	M_o/M_e		M_o/M_e
Vehicle velocity	ft/s	m/s	m/s
Power	hp	W	kW
Length	ft	m	m
Mass	lbm	kg	kg

SI abbreviations

N = newtons
s = seconds
kg = kilograms
m = meter
kW = kilowatts
J = joule
c = centi (10^{-2})
k = kilo (10^3)
M = mega (10^6)

Other units and prefixes are given in E.A. Mechtly, *The International System of Units: Physical Constants and Conversion Factors*, NASA SP-7012, 1973.